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RESEARCH MEMORANDUM

INDIRECT METHODS FOR OBTAINING RAM-JET EXHAUST-GAS TEMPERATURE APPLIED TO FUEL-METERING CONTROL

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SUMMARY

An analytical method has been developed that gives two independent means of obtaining the total-temperature ratio across a ram jet or across a turbojet tail-pipe burner without direct measurement of the final gas temperature. Experimental verification of this analysis has been obtained with a 20-inch ram jet over a wide range of operating conditions. Either method of evaluation described can be used to determine the final gas total temperature of a ram jet or turbojet tail-pipe burner, a temperature that is usually too high for simple direct measurement. Inasmuch as the thrust of a ram jet is dependent upon the value of the total-temperature ratio, these methods of evaluation also provide the basis for two general types of engine fuel-metering control.

INTRODUCTION

In the initial developmental stages of the steady-flow ram jet, the fuel flow to the engine was manually adjusted by means of a throttle in the fuel line in order to allow for variations in altitude, airspeed, and fuel-air ratio. (See references 1 to 3.) Manual control of the fuel flow is expedient for test-stand, wind-tunnel, and flying test-bed studies of the ram jet but is unsatisfactory for piloted-flight application.

The need for a fuel-metering control was recognized at an early stage in the development of the ram jet and several systems have been proposed, developed, and used. The simplest system is a constant fuel-flow control, which is limited in application to an engine designed for constant combustion-chamber conditions and is satisfactory at only the design airspeed and altitude.

Another general type of fuel-metering control being studied, but not yet developed, uses the engine back pressure, which is varied by controlling the fuel flow, to position the shock in the supersonic diffuser. The shock position is detected by a static-pressure wall-orifice survey. Such a control is applicable only at supersonic speeds.

Most of the other proposed fuel-metering controls are designed to maintain a constant preset fuel-air ratio and operate in a manner similar to the carburetors of reciprocating engines. Several of these controls have been developed and at least one has been used successfully in free flight. These devices are limited in use, however, because they neglect the variation in the total-temperature ratio across the ram jet.

References 4 to 6 indicate that at any given airspeed and altitude the net thrust coefficient of a given ram jet is dependent upon the total-temperature ratio across the ram jet and the exhaust-nozzle-throat area. In the design of a ram-jet fuel-metering control, emphasis should therefore be placed not only on the control of the fuel-air ratio, which is but one of the several factors affecting the total-temperature ratio, but on the actual control of the value of the temperature ratio.

If the final gas temperature could be directly measured, it would be a simple matter to evaluate the total-temperature ratio and then control its actual value by varying the fuel flow; however, the temperature values desired for high thrust output and achieved by the gases in a ram jet under most operating conditions are above any known practical means of direct measurement. Until a simple practical means of directly measuring these high temperatures is developed, an indirect method of obtaining the total-temperature ratio must therefore be applied.

An analysis of the process of heat addition in constant-area combustion chambers as applied to ram-jet or turbojet tail-pipe burners indicates two possible means of determining the total-temperature ratio from engine pressures and temperatures that are readily measured: (1) the static-pressure-drop method, which is based on a relation between the total-temperature ratio and the static-pressure-drop coefficient across the ram-jet combustion chamber; and (2) the combustion-chamber-inlet Mach number method, which is based on a relation between the total-temperature ratio and the combustion-chamber-inlet Mach number.

These relations are developed, their experimental verification is presented, and their relative merits are discussed. In addition, several total-temperature-ratio meter designs (hereinafter referred to as "τ-meter") are proposed. These designs provide a simple means of evaluating the total-temperature ratio and also provide the basis for ram-jet fuel-metering controls using the total-temperature ratio as the control variable.

SYMBOLS

The following symbols are used in this analysis:

A	cross-sectional area, square feet
g	acceleration of gravity, feet per second per second
M	Mach number
m	mass flow, slugs per second
P	total pressure, pounds per square foot absolute
p	static pressure, pounds per square foot absolute
Δp	static-pressure drop, pounds per square foot
q	dynamic pressure, pounds per square foot
R	gas constant, foot-pounds per pound per $^{\circ}F$
T	total temperature, $^{\circ}R$
t	static temperature, $^{\circ}R$
V	velocity, feet per second
γ	ratio of specific heat at constant pressure to specific heat at constant volume
μ	ratio of mass flow at combustion-chamber outlet m_5 to mass flow at combustion-chamber inlet m_3
ρ	density, slugs per cubic foot
τ	total-temperature ratio across ram jet, T_6/T_x (where T_x is taken as equal to T_0)

a, a'
B, B', B''
b, b'
C
c
K

constants

Subscripts:

0	free stream
1	supersonic-diffuser inlet
2	supersonic-diffuser throat
3	diffuser exit and combustion-chamber inlet
4	station immediately downstream of burner
5	combustion-chamber outlet
6	exhaust-nozzle throat
x	between stations 2 and 3 but upstream of point of fuel injection

APPARATUS AND PROCEDURE

The experimental data presented were obtained in an investigation of a typical 20-inch ram jet in the Cleveland altitude wind tunnel. The engine was mounted in the wind-tunnel test section below a 7-foot chord wing, which was supported at the tips by the wind-tunnel balance frame. Dry refrigerated air was supplied to the ram jet through a pipe from the wind-tunnel make-up air duct. This air was available in the make-up air duct at approximately sea-level pressure and was throttled to provide the desired total pressure at the diffuser inlet. The ram jet exhausted directly into the wind tunnel where the static pressure was varied to obtain different values of ram-pressure ratio across the engine.

This installation eliminates the need for a supersonic diffuser; the subsonic diffuser used in this investigation began at station x (fig. 1). For data presented as a function of free-stream Mach number M_0, an equivalent M_0 calculated from a pressure ratio is used. For values of M_0 less than 1, the subsonic-diffuser-inlet total pressure was taken as the free-stream total pressure. For values of M_0 greater than 1, the measured diffuser-inlet total pressure was adjusted in accordance with the ratio of the total pressures across an assumed supersonic diffuser. This pressure ratio was taken as the theoretical pressure ratio across a normal shock at the throat of a convergent-divergent supersonic diffuser designed to allow shock entry at the pertinent M_0 value.

Restraint of the model by the ram pipe was obviated by a sealed slip joint inserted between the ram pipe and the diffuser inlet. The tunnel balance system could then be used to measure the engine thrust. The value of the final gas total temperature was computed from values of engine thrust and gas flow (references 5 and 6). A survey rake was used to determine the diffuser-inlet total pressure P_x, the diffuser-inlet static pressure p_x, and the diffuser-inlet total temperature T_x. These pressures and temperatures were used to compute the air flow through the engine. The combustion-chamber-inlet static pressure p_3 was taken as the average of five static-pressure wall-orifice readings; the static pressure immediately downstream of the burner p_4 was obtained with a single static-pressure wall orifice, and the combustion-chamber-outlet static pressure p_5 was taken as the average of seven static-pressure wall-orifice readings. A rotameter was used to measure the fuel flow to the engine.

The range of operation covered by the data presented includes equivalent free-stream Mach numbers M_0 from 0.64 to 1.44, combustion-chamber-inlet Mach numbers M_3 from 0.09 to 0.18, pressure altitudes from 10,000 to 35,000 feet, fuel-air ratios from 0.028 to 0.065, and combustion efficiencies from 52 to 79 percent. The total temperature T_x was held at ±10° F and the fuel-injection temperature was held at 200° ±10° F.

DETERMINATION OF TOTAL-TEMPERATURE RATIO BY STATIC-PRESSURE-DROP METHOD

Analysis

In applying the static-pressure-drop method of determining the total-temperature ratio to the problem of obtaining ram-jet exhaust-gas temperatures, a general expression relating the conditions before and after heat addition to a nonviscous compressible fluid flowing in a constant-area tube is used:

$$\mu\left(\frac{R_5\ T_5}{R_3\ T_3}\right)^{1/2} = \frac{M_5}{M_3}\left(\frac{\gamma_5}{\gamma_3}\right)^{1/2}\frac{1+\gamma_3\ M_3{}^2}{1+\gamma_5\ M_5{}^2}\left(\frac{1+\frac{\gamma_5-1}{2}\ M_5{}^2}{1+\frac{\gamma_3-1}{2}\ M_3{}^2}\right)^{1/2} \qquad (1)$$

This equation is based on one-dimensional-flow relations and is derived in the appendix as equation (A9).

The solution of the square of equation (1) is plotted in figure 2, which gives the variation of the combustion-chamber-outlet Mach number M_5 with the combustion-chamber-inlet Mach number M_3 and the quantity $\mu^2\left(\frac{R_5\ T_5}{R_3\ T_3}\right)$ for $\gamma_3 = 1.4$ and $\gamma_5 = 1.3$. These assumed values approximate those encountered in actual engine operation. The curves in figure 2 show the change in the Mach number of gases flowing through a constant-area tube caused by the addition of heat to the gases. Turbulence losses introduced by a flame holder, which have been included in similar data presented in reference 7, are not included here. It can be assumed for simplicity that all the fuel has already been introduced at station 3; then $\mu = 1$. In addition, R_5/R_3 can be assumed equal to 1 with little error. If these assumptions are made, the curves of figure 2 give directly the variation in M_5 with changes in T_5/T_3 and M_3.

An expression for the static-pressure drop in gases flowing through a constant-area tube resulting from the addition of heat is given by the following equation in terms of dynamic pressure at the tube inlet and Mach number and ratio of specific heats γ at the tube inlet and outlet:

$$\frac{p_3 - p_5}{2q_3} = \left(\frac{1+\gamma_3\ M_3{}^2}{1+\gamma_5\ M_5{}^2}\right)\left(\frac{M_5{}^2\ \gamma_5}{M_3{}^2\ \gamma_3}\right) - 1 \qquad (2)$$

This expression is derived in the appendix as equation (A11).

When data from figure 2 are substituted into equation (2) and the expression solved, the variation of $\frac{p_3 - p_5}{2q_3}$ with M_3 and $\mu^2\left(\frac{R_5}{R_3}\frac{T_5}{T_3}\right)$ for $\gamma_3 = 1.4$ and $\gamma_5 = 1.3$ can be obtained (fig. 3).

A similar curve in terms of the total-pressure loss is given in reference 8. The choking line in figure 3 is the condition at which M_5 becomes equal to 1.0.

The data of figure 3 have been cross-plotted in figure 4 in order to show the variation between $\frac{p_3 - p_5}{2q_3}$ and $\mu^2 \left(\frac{R_5}{R_3} \frac{T_5}{T_3}\right)$ at constant values of M_3. If $\mu^2 \frac{R_5}{R_3}$ is assumed to be equal to 1, at a given value of M_3 the curves of figure 4 indicate an approximately linear relation between $\frac{p_3 - p_5}{2q_3}$ and T_5/T_3 over a limited range of variation of T_5/T_3. The range of variation of T_5/T_3 over which this approximation holds decreases at larger values of M_3. If M_3 is small, q_3 can be expressed as $P_3 - p_3$, and when the assumption of linear variation between $\frac{p_3 - p_5}{2q_3}$ and T_5/T_3 is applied, equation (2) can be approximated by the general form

$$\frac{p_3 - p_5}{P_3 - p_3} = a' \left(\frac{T_5}{T_3}\right) + b' \qquad (3)$$

where a' and b' are constants peculiar to the configuration and the range of T_5/T_3 and M_3.

An expression of the type given by equation (3) can be derived for the total-pressure loss across a constant-area combustion chamber (reference 9) and used to evaluate the ram-jet exhaust-gas temperature (reference 10); however, the actual measurement of the total pressure in a high-velocity, high-temperature gas stream is difficult, whereas the static pressure can be measured easily with a static wall orifice.

Equation (3) is an expression relating the conditions before and after heat addition in a combustion chamber in which friction has been neglected. In an actual combustion chamber, the over-all static-pressure loss can be measured and divided, as suggested in reference 9, into combustion losses and friction losses. Because the effect of any fuel addition after station 3 is negligible, the static-pressure drop between stations 3 and 5 can then be given in terms of the dynamic pressure at station 3 as

$$\frac{\Delta p_{3-5}}{q_3} = \left(\frac{\Delta p_{3-4}}{q_3}_{\text{burner}} + \frac{\Delta p_{4-5}}{q_3}_{\text{shell}}\right)_{\text{friction}} + \left(\frac{\Delta p_{3-5}}{q_3}\right)_{\text{combustion}} \tag{4}$$

Because the friction static-pressure-drop coefficient for a given engine configuration remains, in practice, essentially constant with the changes in heat addition usually encountered, a measurement of the over-all static-pressure-drop coefficient $\frac{\Delta p_{3-5}}{q_3}$ can be used to give a measure of the combustion static-pressure-drop coefficient $\left(\frac{\Delta p_{3-5}}{q_3}\right)_{\text{combustion}}$. The general form of equation (3) may thus be applied to the over-all static-pressure drop instead of to merely the combustion static-pressure drop, which is difficult to evaluate in practice. In the discussion that follows, (p_3-p_5) therefore refers to the over-all static-pressure drop across the combustion chamber.

If fuel is introduced in the diffuser upstream of station 3 in order to allow premixing of the air and the fuel before combustion, it may be desirable, in using equation (3) for evaluating T_5/T_3, to measure conditions at station 3 by other means than conventional total- and static-pressure tubes, which may become plugged with fuel. Of several methods available, an indirect means of obtaining the value of the quantity P_3-p_3 was chosen for this investigation. From the expression for the conservation of mass (neglecting the change in mass due to fuel addition) and within the incompressible-flow restrictions, it can be shown that

$$P_3 - p_3 = (P_x-p_x)\left(\frac{A_x}{A_3}\right)^2 \frac{p_x}{p_3}\frac{t_3}{t_x} \tag{5}$$

Station x was chosen in the diffuser upstream of the point of fuel injection, but downstream of the assumed shock at station 2. The actual value of p_3 was determined with a static wall orifice. The experimental data indicate that for a 20-inch ram jet and a value of A_x/A_3 of 0.5 the variation in the quantity

$$\left(\frac{A_x}{A_3}\right)^2 \frac{p_x}{p_3}\frac{t_3}{t_x}$$

was very slight over the entire range at which the ram jet was operated, even with preheated fuel injected upstream of station 3. Equation (5) can therefore be reduced to

$$P_3 - p_3 = c(P_x - p_x) \tag{6}$$

where c is a constant expressing the quantity

$$\left(\frac{A_x}{A_3}\right)^2 \frac{p_x}{p_3} \frac{t_3}{t_x}$$

If equation (6) is substituted into equation (3), the relation expressed by equation (3) can be given in terms of conditions at station x

$$\frac{p_3 - p_5}{P_x - p_x} = a\left(\frac{T_5}{T_3}\right) + b \tag{7}$$

The total-temperature ratio across the engine T_6/T_x is more nearly a measure of engine performance than T_5/T_3. If the temperature rise in the exhaust nozzle is assumed to be negligible, T_6 can be substituted for T_5. In addition, the temperature difference between the point of fuel injection and the combustion-chamber inlet may be assumed to be negligible and T_x substituted for T_3. Equation (7) then becomes

$$\frac{p_3 - p_5}{P_x - p_x} = a\left(\frac{T_6}{T_x}\right) + b = a \cdot \tau + b \tag{8}$$

If the constants a and b are known, the measurements required for the calculation of T_6 from equation (8) are simple. The temperature T_x can be measured with a thermocouple or an expansion or resistance-bulb thermometer. The total pressure P_x can be measured with a total-pressure tube. The static pressure p_x can be measured with either a static-pressure tube or a static wall orifice and static pressures p_3 and p_5 can be measured with static wall orifices. The effect of the flame-holder pressure drop can be eliminated by using a static pressure p_4 measured immediately downstream of the flame holder instead of p_3.

Experimental Verification

Experimental data obtained during altitude-wind-tunnel investigations of a typical 20-inch ram-jet configuration are presented in figure 5 in the form suggested by equation (8). The over-all static-pressure-drop coefficient across the combustion chamber is presented in this figure as a function of the total-temperature ratio across the engine T_6/T_x. The experimental data indicate that the substitution of T_6/T_x for T_5/T_3 in equation (7) can be made successfully.

Also given in figure 5 are the values of combustion-chamber-inlet Mach number M_3 for representative data points, and the static-pressure-drop coefficient across the flame holder as a function of the total-temperature ratio across the ram jet. The data indicate that the flame-holder static-pressure-drop coefficient is constant and does not vary with τ.

The data in figure 5(a) were obtained with a 20-inch ram jet having an 8-foot combustion chamber and a 2-foot converging exhaust nozzle with a 15-inch exhaust-exit diameter. Data for the same combustion-chamber length with converging nozzles having 17- and $18\frac{3}{4}$-inch-exit diameters are presented in figures 5(b) and 5(c), respectively. Data for a 5-foot combustion chamber with 15- and 17-inch exhaust-nozzle exit diameters are presented in figures 5(d), and 5(e), respectively.

The experimental data substantiate, with little scatter, the trends indicated by the theory and show that the general form of equation (8) can be applied to any of the configurations. The mean curve drawn through all the data of figure 5(a) is given by the simple linear expression

$$\frac{p_3 - p_5}{P_x - p_x} = 0.762\left(\frac{T_6}{T_x}\right) - 0.50 \qquad (9)$$

and is independent of actual values of altitude, airspeed, fuel-air ratio, and combustion efficiency. The same curve has been drawn through the data of figures 5(b) and 5(c) and, as in figure 5(a), there is little scatter of the experimental data. The same expression (equation (9)) has therefore been used to give the relation between the over-all static-pressure-drop coefficient across a given combustion chamber and the total-temperature ratio for a variation in exhaust-nozzle area of 56 to 88 percent of the combustion-chamber area.

The apparent insensitivity of the constants in equation (9) to large changes in the area ratio between the combustion chamber and the exhaust nozzle is a desirable feature because it will permit the use of the general form of equation (8) with ram jets having variable-area exhaust nozzles.

Variation in the internal friction losses will cause a change in the constants a and b of equation (8). The changes in these constants effected by shortening the combustion chamber is indicated by a comparison of figures 5(a), 5(b), and 5(c) for an 8-foot combustion chamber and figures 5(d) and 5(e) for a 5-foot combustion chamber. The relative insensitivity of the constants in equation (8) to nozzle diameter is again demonstrated in figures 5(d) and 5(e); the same linear curve has been drawn on both figures. In this case, however, the curve is given by the expression

$$\frac{p_3 - p_5}{P_x - p_x} = 0.682\left(\frac{T_6}{T_x}\right) - 0.60 \qquad (10)$$

and differs from equation (9) only because of the decreased shell-friction pressure drop.

During operation some heat will be lost through the combustion-chamber walls to the ambient atmosphere or to a cooling medium; however, the relations developed are still valid and, furthermore, within the restrictions of one-dimensional flow, these relations can be applied to constant-area combustion chambers of arbitrary shapes.

In addition to the application to a ram jet, a preliminary study of experimental data indicates that the analysis presented can also be applied to a constant-area tail-pipe burner on a turbojet engine, which might be expected because the tail-pipe burner on a turbojet engine is essentially a ram jet.

The analytical development that resulted in equations (1) and (2) assumed that heat was being added to gases flowing through a constant-area combustion chamber. On some ram jets, heat may be added to the gases in a diverging or converging combustion chamber. The development of a general expression for the process of heat addition in such a combustion chamber is difficult because the manner and rate at which the heat addition is accomplished determines the pressure and Mach number changes. The nearest approach to a general expression that can be given for this type of process is equation (A6). The simple method of obtaining τ from the static-pressure drop across the combustion chamber outlined herein cannot be directly used for heat addition in a nonuniform-area combustion chamber.

Based upon the success achieved in replacing a nonuniform-area combustion chamber with an equivalent constant-area chamber for purposes of calculation, as was done in reference 9, a linear relation may also result between τ and the pressure drop in a nonuniform-area combustion chamber.

Application

The relation between the static-pressure-drop coefficient and τ has immediate application in the design of: (1) a τ-meter and a final-gas-temperature indicator, and (2) a fuel-metering control to permit the presetting of τ before take-off or, if desired, its variation in flight.

Two general τ-meter designs, which also provide the basis for a fuel-metering control, are proposed. The first design is mechanical; the second design utilizes an electrical circuit to indicate the value of τ. These designs can be used at both subsonic and supersonic flight velocities. From the fuel-metering aspect, the designs are developed only to the point where an indication is given if a change in fuel flow is required and whether the change should be an increase or a decrease in fuel flow. The mechanism for actually changing the fuel flow is a detail of the fuel system used and will not be discussed.

The over-all static-pressure loss can be expressed, as previously indicated, in the form of equation (8) For an application in which τ is constant, equation (8) can be reduced to an even simpler form as

$$p_3 - p_5 = C(P_x - p_x) \tag{11}$$

where C is a constant replacing the term $\left(a \cdot \tau + b\right)$.

Expressions such as equations (8) and (11) may be applied to τ-meters utilizing a multiple diaphragm or a bellows system. Such a meter provides the basis for a fuel-metering control. A schematic diagram of a τ-meter that utilizes a bellows system and is based on mechanical principles is illustrated in figure 6(a). The position of the fulcrum is fixed by the constant C in equation (11). If diaphragms are used instead of bellows, a fixed fulcrum position can be replaced by a series of diaphragms of given area ratios. If the ram jet is operated over a range of τ values, the fulcrum position can be made adjustable and calibrated in terms of τ according to equation (8).

A needle pivoting with the balance arm about the fulcrum indicates whether the system is balanced and whether the engine is operating at the desired value of τ. If the system is out of balance, the needle will indicate whether an increase or decrease in τ (and consequently in fuel flow) is required to obtain balance.

The schematic diagram of a τ-meter using a Wheatstone bridge circuit, also based on equation (8), is shown in figure 6(b). The actual pressure differences can be obtained with bellows or diaphragms and are made to vary a resistance in the circuit that is proportional to the pressure difference. When used as a τ-meter, the resistance $a \cdot \tau$ in figure 6(b) is varied until the bridge is balanced. This resistance is calibrated in terms of τ and the value of τ is thus determined. When used in a control system, the meter will show the amount and direction of τ unbalance. This indication can then be used to cause a variation in the fuel flow as required until a balance in the bridge is restored, thus indicating that the value of τ desired has been obtained.

A variation in the exhaust-nozzle area at a given value of M_0 and τ will also cause a variation in the net thrust of the engine. If these suggested fuel-metering systems are used with ram jets having adjustable-area exit nozzles, a thrust calibration in terms of A_4 as well as M_0 and τ is therefore required.

DETERMINATION OF TOTAL-TEMPERATURE RATIO BY MACH NUMBER METHOD

Analysis

The Mach number method of determining τ differs from the static-pressure-drop method in that the Mach number method uses only a measure of the combustion-chamber-inlet dynamic and static pressures in evaluating τ. In applying this method to the determination of τ, an expression based on the conservation of mass that relates conditions at the combustion-chamber inlet to those at the exhaust-nozzle throat is used. This relation is given by the following equation:

$$\mu \frac{p_3}{p_6} \frac{A_3}{A_6} \left(\frac{R_6}{R_3} \frac{T_6}{T_3}\right)^{1/2} = \frac{M_6}{M_3}\left(\frac{\gamma_6}{\gamma_3}\right)^{1/2} \left(\frac{1 + \frac{\gamma_6 - 1}{2} M_6^2}{1 + \frac{\gamma_3 - 1}{2} M_3^2}\right)^{1/2} \tag{12}$$

An expression similar to this equation is derived in the appendix as equation (A6).

Several simplifying assumptions are required before equation (12) can be reduced to an expression suitable for application to experimental data. Equation (12) can also be given as

$$M_3 \sqrt{\frac{T_6}{T_3}} = K \frac{A_6}{A_3} \frac{p_6}{P_3} M_6 \left(1 + \frac{\gamma_6 - 1}{2} M_6^2\right)^{1/2} \tag{13}$$

where K is the term

$$\left[\frac{1}{\mu}\left(\frac{R_3}{R_6} \frac{\gamma_6}{\gamma_3}\right)^{1/2} \left(1 + \frac{\gamma_3 - 1}{2} M_3^2\right)^{\frac{\gamma_3 + 1}{2(\gamma_3 - 1)}}\right]$$

If the quantity $\frac{1}{\mu}\left(\frac{R_3}{R_6} \frac{\gamma_6}{\gamma_3}\right)^{1/2}$ is assumed constant and if M_3 is sufficiently small that second order terms of M_3 can be neglected, then the value of K in equation (13) can be assumed constant. If the right side of equation (13) is multiplied and divided by P_6 and if the general expression relating the total and static pressures and the Mach number is applied, equation (13) becomes

$$M_3 \sqrt{\frac{T_6}{T_3}} = K \frac{A_6}{A_3} \frac{P_6}{P_3} \frac{M_6}{\left(1 + \frac{\gamma_6 - 1}{2} M_6^2\right)^{\frac{(\gamma_6 + 1)}{2(\gamma_6 - 1)}}} \tag{14}$$

The relation given by equation (14) indicates that for a given area ratio between the exhaust-nozzle throat and the combustion-chamber inlet, the parameter $M_3 \sqrt{\frac{T_6}{T_3}}$ is a function of M_6 and P_6/P_3. Before choking occurs at station 6, the quantity $M_3 \sqrt{\frac{T_6}{T_3}}$

will increase as M_6 increases if the ratio P_6/P_3 is held constant. After choking occurs at station 6, M_6 becomes and remains approximately equal to 1.0; if the ratio P_6/P_3 is fixed, the quantity $M_3\sqrt{\frac{T_6}{T_3}}$ therefore also becomes constant. For convenience, $M_3\sqrt{\frac{T_6}{T_3}}$ can be expressed as a function of M_0, inasmuch as M_6 is a function of M_0 (reference 5). After the engine is choked, however, M_6 and $M_3\sqrt{\frac{T_6}{T_3}}$ become constant and do not vary with M_0.

Experimental Verification

The variation of $M_3\sqrt{\frac{T_6}{T_x}}$ with M_0 for a typical 20-inch ram jet with a 15-inch and a 17-inch diameter exhaust nozzle is given in figure 7. The approximate pressure altitudes at which the data were obtained are indicated. As with the data presented for the static-pressure-drop method of determining τ, T_x is assumed equal to T_3.

The condition at which $M_3\sqrt{\frac{T_6}{T_x}}$ becomes constant is shown in figure 7. At a fixed-area ratio A_6/A_3, any changes in the pressure drop between stations 3 and 6 caused by a variation in the fuel-injector, flame-holder, or combustion-chamber length, will displace the curve. The large shift in the curve caused by changing the exhaust-nozzle diameter from 15 to 17 inches is due primarily to the change in area ratio A_6/A_3, and secondarily, to an accompanying change in the pressure ratio P_6/P_3 (reference 6). It might be expected that for a given exhaust-nozzle diameter, variations in pressure loss caused by changes in τ would scatter the data; however, the curves indicate that over the range of τ and M_3 investigated this effect is negligible.

Also presented in figure 7 are two theoretical curves based on equation (14) of this report and equation (13) of reference 5 for a 20-inch ram jet with 15- and 17-inch exhaust-nozzle diameters. It was assumed that the second-order terms of M_3 can be neglected and that $P_6/P_3 = 1$, $\mu = 1$, $R_3 = 53.3$, $R_6 = 53.8$,

$\gamma_3 = 1.4$, and $\gamma_6 = 1.3$. The displacement of the experimental data from these theoretical curves is due primarily to the total-pressure losses encountered in the actual engine.

For any given engine for which a curve similar to those shown in figure 7 is available, the value of T_6 or τ can be easily determined. In order to obtain the value of T_6, only the values of M_3, T_x, and M_0 need be known. If the value of τ is desired, only M_3 and M_0 need be measured. Once the ram jet is choked, the actual value of M_0 is not needed to determine either T_6 or τ.

Application

The relation between τ, M_3, and M_0 has four applications to ram jets of a given design and geometry:

(1) A τ-meter and final-gas-temperature indicator

(2) Fuel-metering control to permit presetting of τ before take-off or its variation in flight

(3) A combustion-chamber-inlet Mach number control to limit or vary M_3 by controlling τ through changes in fuel flow

(4) A supersonic-diffuser shock-positioning control to operate by controlling M_3 at the pertinent value for a given M_0

The fourth application will not be discussed.

As with the static-pressure-drop method of determining τ, two general τ-meter designs are presented that also provide the basis for a fuel-metering control. The first design is mechanical, whereas the second design uses an electrical circuit to indicate values of τ. These designs are for supersonic-flight applications only. From the fuel-metering aspect, the designs are again developed to the point where an indication is given if a change in fuel flow is required and the mechanism of actually changing the fuel flow will not be discussed.

The supersonic portion of the curves on figure 7 can be expressed as

$$M_3\sqrt{\frac{T_6}{T_x}} = B'' \tag{15}$$

Because the velocities encountered at station 3 in ram jets are low enough to neglect compressibility effects, M_3 can be given with reasonable accuracy by $\sqrt{\frac{2(P_3-p_3)}{\gamma_3 \, p_3}}$. Assuming a constant value of γ_3, equation (15) becomes

$$\frac{P_3-p_3}{p_3} = \frac{B'}{\tau} \tag{16}$$

where $B' = \frac{\gamma_3 (B'')^2}{2}$

For a constant value of τ, equation (16) reduces to

$$\frac{P_3-p_3}{p_3} = B \tag{17}$$

where B equals $\frac{B'}{\tau}$.

As previously discussed, (P_3-p_3) can be expressed in terms of (P_x-p_x) if measurements at station 3 are difficult to obtain. (See equation (6).)

The schematic diagrams of τ-meters based on equation (16) are shown in figure 8. Figure 8(a) presents a design using mechanical principles and figure 8(b) a design using a Wheatstone bridge circuit. Both units must be operated in a sealed container maintained at a constant internal pressure in order to eliminate the effect of variable ambient pressure on the unbalanced bellows. The operating principles and techniques of these designs are the same as those discussed for the τ-meter designs presented for the static-pressure-drop method of determining τ. No modification of these designs is required for their use in limiting combustion-chamber-inlet Mach number. It requires only that a minimum value of τ be imposed, thus limiting the maximum M_3 that will occur.

CONCLUSION

Two effective means are available for determining the total-temperature ratio τ across a ram jet or across a turbojet tail-pipe burner, as well as the final gas temperature when simple direct temperature measurement is not possible. The first method is based on a relation between the total-temperature ratio and the static-pressure drop across the combustion chamber. Only fixed combustion-chamber geometry and configuration are required and variation in exhaust-nozzle-exit area is tolerable. There is no discontinuity in the relation at choking and it is independent of actual values of altitude, airspeed, fuel-air ratio, and combustion efficiency.

The second method is based on a relation between the total-temperature ratio and the combustion-chamber-inlet Mach number and the free-stream Mach number. This relation requires fixed engine geometry and fixed internal frictional losses and is discontinuous at choking.

Additional τ-meter designs undoubtedly can be conceived that use mechanical or electrical devices, or a combination of both, actuated by pressures directly or by pressure differences. The purpose of this paper, however, is not to describe all the possible devices, but to indicate that both the aforementioned relations can be used to evaluate τ. Inasmuch as most current turbojet tail-pipe-burner designs have constant-area combustion chambers, the methods of τ evaluation and fuel-metering control suggested can be applied to turbojet engines with tail-pipe burning as well as to ram jets.

Some precautions must be observed in the use of τ as a control variable. First, an overriding control is required to limit the fuel flow to that value at which maximum τ occurs. If the fuel-metering control calls for an increase in τ after the maximum τ has been reached, further increases in fuel flow may cause τ to decrease and continue to decrease as the fuel flow is increased.

Secondly, the fuel-air-ratio range at which a ram jet is operated may sometimes be fixed by the operational limits of the configuration rather than by the maximum τ value attainable. In such cases the overriding control may be set for a limiting fuel-flow range instead of merely a maximum fuel flow.

An additional restriction of a maximum final gas temperature may also be imposed on the control. Such a restriction results from materials and cooling limitations. With this restriction, the τ-measuring element of the control can be used to indicate the final gas temperature and a temperature-limiting feature can be added to the overriding control.

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APPENDIX - THEORETICAL ANALYSIS

In order to provide the basis for a study of the temperature increase in the gases flowing through a ram jet and to provide methods of evaluating this increase without direct temperature measurement, a theoretical analysis must be made to determine the effect of heat addition to gases flowing in a constant-area tube. The type of supersonic diffuser used is independent of the analysis to be developed. The analysis assumes a nonviscous one-dimensional compressible fluid and is presented in a manner that is most useful for the applications discussed herein.

The expression for the conservation of mass between stations 3 and 5, the combustion chamber, (fig. 1) is given as

$$\rho_3 A_3 V_3 = \rho_5 A_5 V_5 \tag{A1a}$$

If fuel is added after station 3, equation (A1a) becomes

$$\rho_3 A_3 V_3 \left(\frac{m_5}{m_3}\right) = \rho_5 A_5 V_5 \tag{A1b}$$

But

$$V = M \sqrt{\gamma g R t} \tag{A2}$$

and

$$\rho = \frac{p}{gRt} \tag{A3}$$

By combining equations (A1b), (A2), and (A3), the following relation is obtained

$$\frac{p_3}{p_5} = \frac{A_5}{A_3} \frac{M_5}{M_3} \left(\frac{\gamma_5}{\gamma_3}\right)^{1/2} \left(\frac{R_3 t_3}{R_5 t_5}\right)^{1/2} \left(\frac{1}{\mu}\right) \tag{A4}$$

where

$$\mu = \frac{m_5}{m_3}$$

Inasmuch as

$$t = \frac{T}{1 + \frac{\gamma-1}{2} M^2} \tag{A5}$$

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equation (A4) can therefore be written as

$$\mu \frac{p_3}{p_5} \frac{A_3}{A_5} \left(\frac{R_5\ T_5}{R_3\ T_3}\right)^{1/2} = \frac{M_5}{M_3} \left(\frac{\gamma_5}{\gamma_3}\right)^{1/2} \left(\frac{1 + \frac{\gamma_5-1}{2} M_5^2}{1 + \frac{\gamma_3-1}{2} M_3^2}\right)^{1/2} \tag{A6}$$

Because $A_3 = A_5$, if any momentum effects due to fuel addition after station 3 are neglected, the conservation of momentum between stations 3 and 5 may be given as

$$p_3 + \rho_3\ V_3^2 = p_5 + \rho_5\ V_5^2 \tag{A7}$$

which becomes, in terms of Mach number,

$$\frac{p_3}{p_5} = \frac{1 + \gamma_5\ M_5^2}{1 + \gamma_3\ M_3^2} \tag{A8}$$

Substituting equation (A8) into equation (A6) gives the following expression:

$$\mu \left(\frac{R_5\ T_5}{R_3\ T_3}\right)^{1/2} = \frac{M_5}{M_3} \left(\frac{\gamma_5}{\gamma_3}\right)^{1/2} \frac{1 + \gamma_3\ M_3^2}{1 + \gamma_5\ M_5^2} \left(\frac{1 + \frac{\gamma_5-1}{2} M_5^2}{1 + \frac{\gamma_3-1}{2} M_3^2}\right)^{1/2} \tag{A9}$$

Equation (A9) gives the variation of Mach number across a constant-area combustion tube due only to heat addition and does not include any pressure losses caused by shell friction or by the burner.

By rearranging the terms of equation (A7), the following equation is obtained:

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$$\frac{p_3 - p_5}{\rho_3 V_3^{\ 2}} = \frac{\rho_5 V_5^{\ 2}}{\rho_3 V_3^{\ 2}} - 1 \tag{A10}$$

Then, by substituting from equations (A2), (A3), and (A8), equation (A10) can be given in terms of Mach number as

$$\frac{p_3 - p_5}{2q_3} = \left(\frac{1 + \gamma_3 M_3^{\ 2}}{1 + \gamma_5 M_5^{\ 2}}\right)\left(\frac{M_5^{\ 2}\ \gamma_5}{M_3^{\ 2}\ \gamma_3}\right) - 1 \tag{A11}$$

Equation (A11) is an expression for the static-pressure-drop coefficient in a constant-area tube, due only to heat addition. A similar expression can also be derived, if desired, for the total-pressure-drop coefficient. (See reference 8.)

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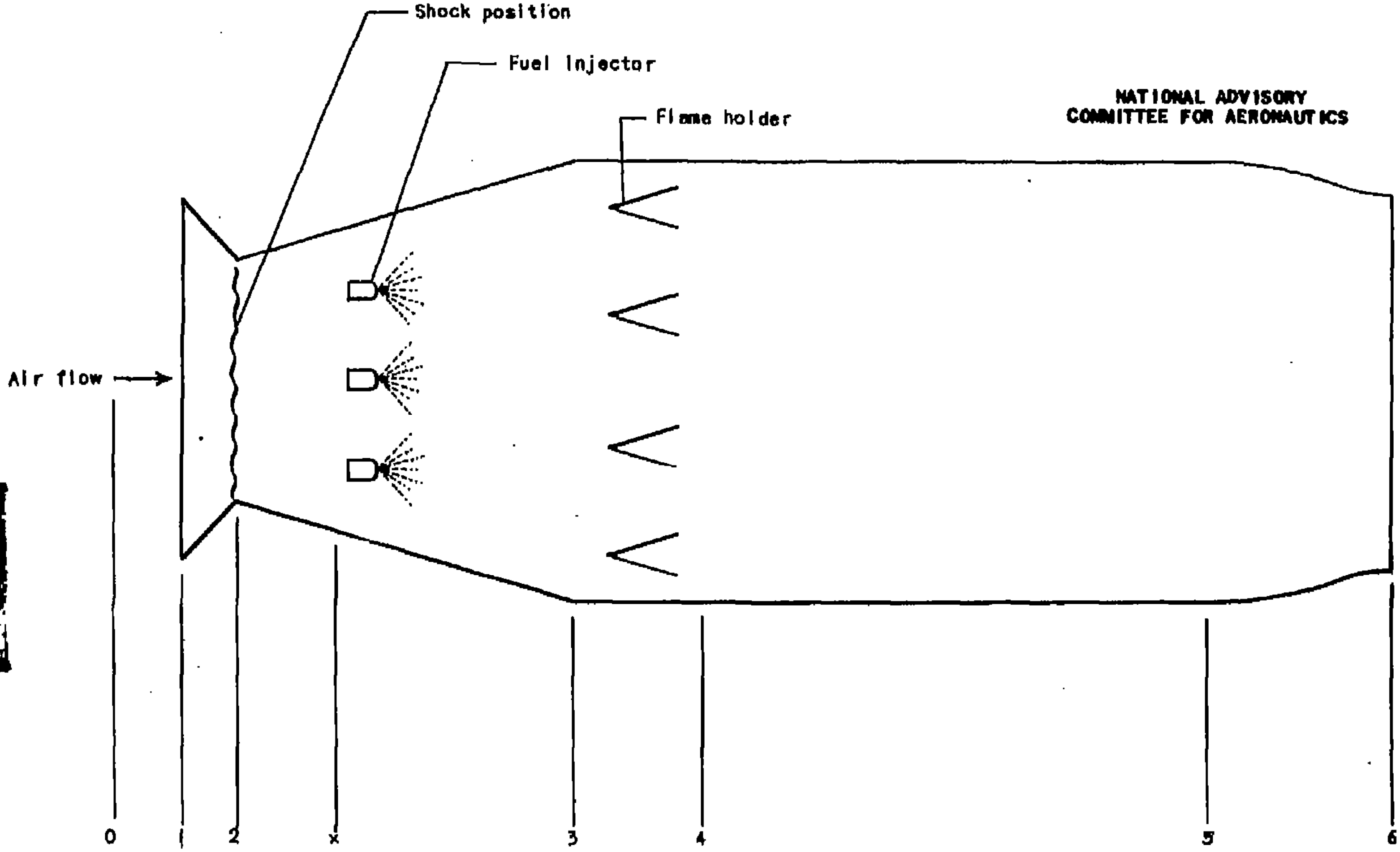

Figure 1. - Schematic diagram of conventional ram jet.

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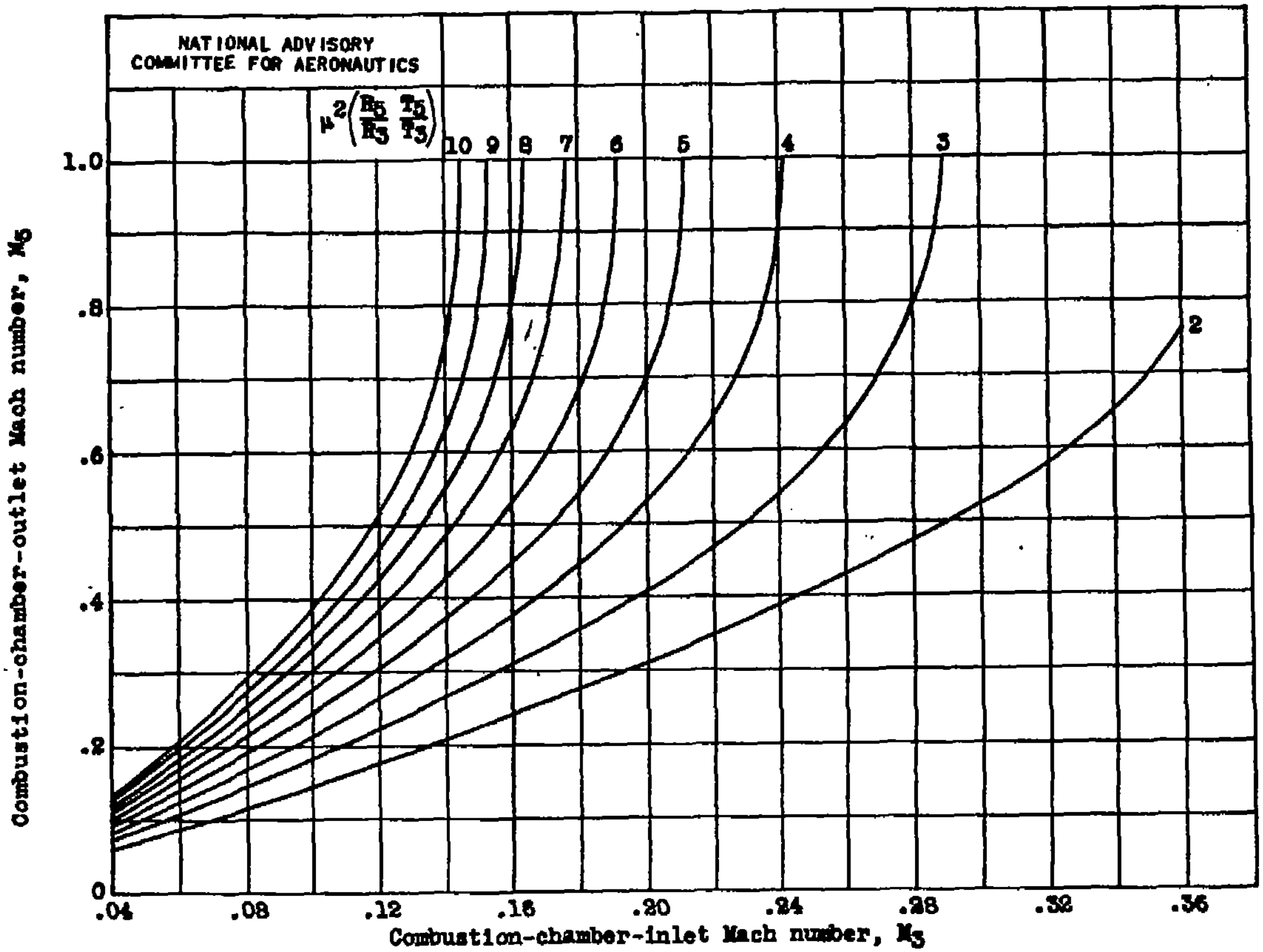

Figure 2. - Theoretical variation of combustion-chamber-outlet Mach number M_5 with combustion-chamber-inlet Mach number M_3 for various values of $\mu^2\left(\frac{R_5}{R_3}\frac{T_5}{T_3}\right)$ with $\gamma_3 = 1.4$ and $\gamma_5 = 1.3$.

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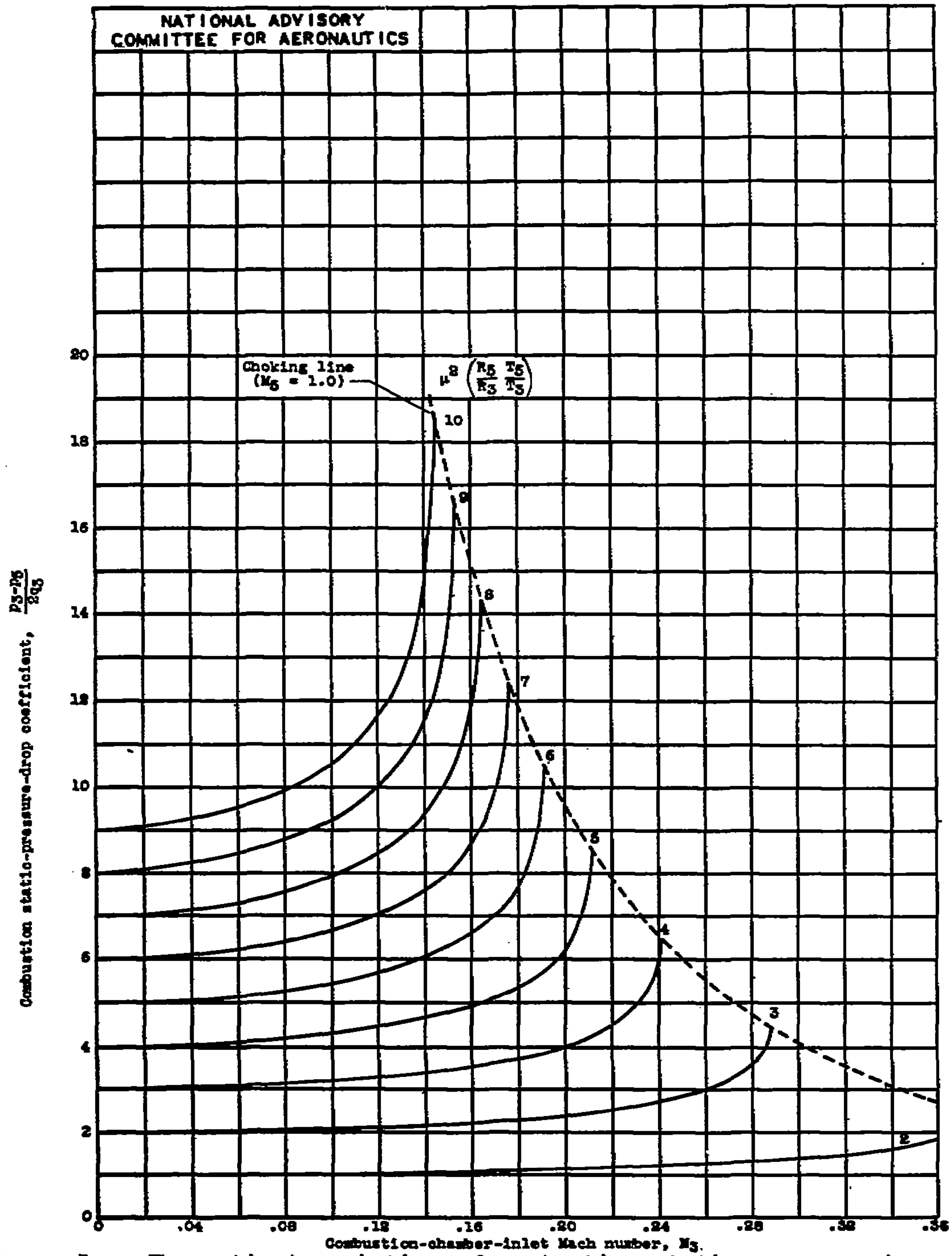


Figure 3. - Theoretical variation of combustion static-pressure-drop coefficient $\frac{p_3 - p_5}{2q_3}$ with combustion-chamber-inlet Mach number M_3 for various values of $\mu^2\left(\frac{R_5}{R_3}\frac{T_5}{T_3}\right)$ with $\gamma_3 = 1.4$ and $\gamma_5 = 1.3$.

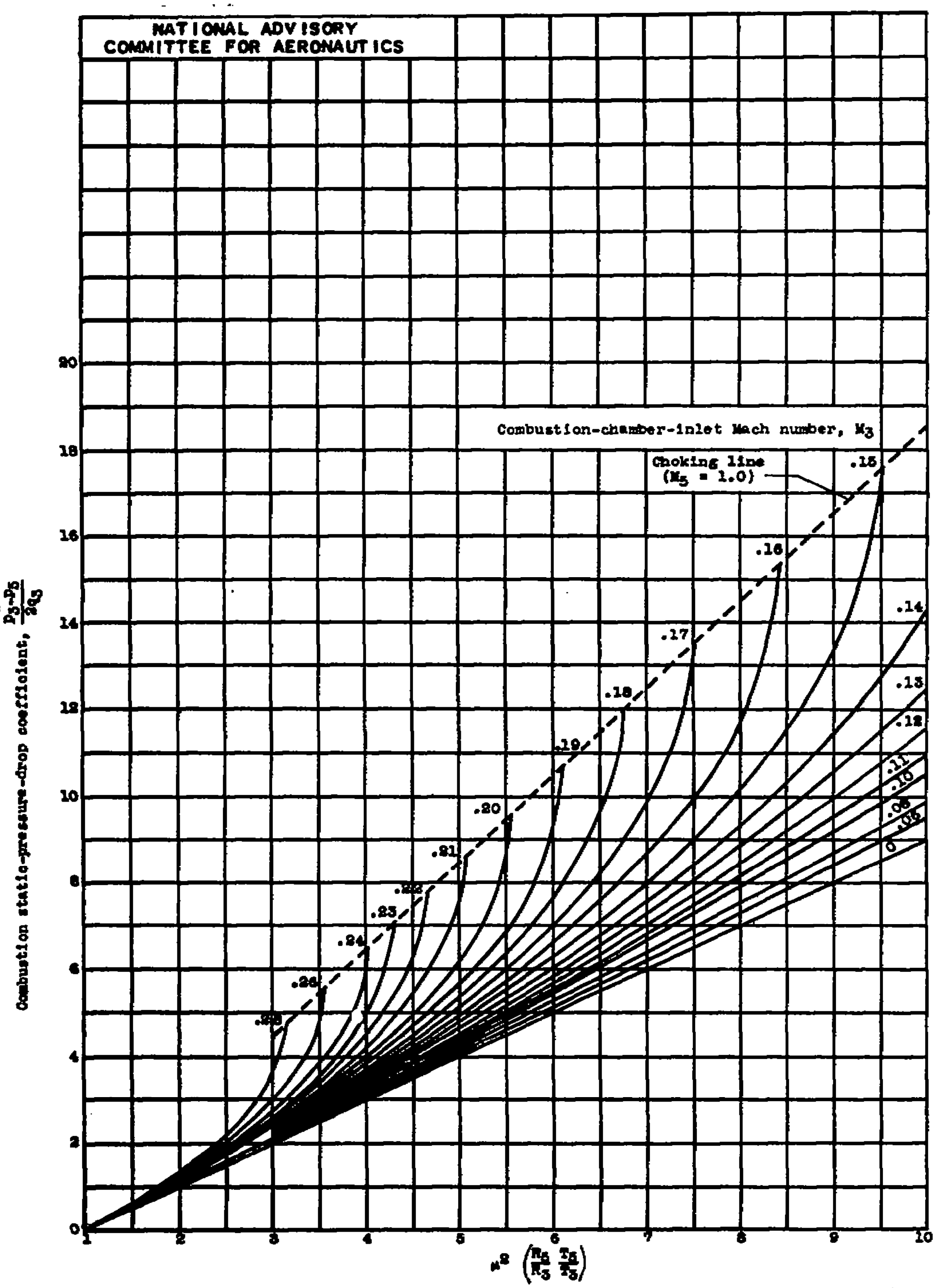
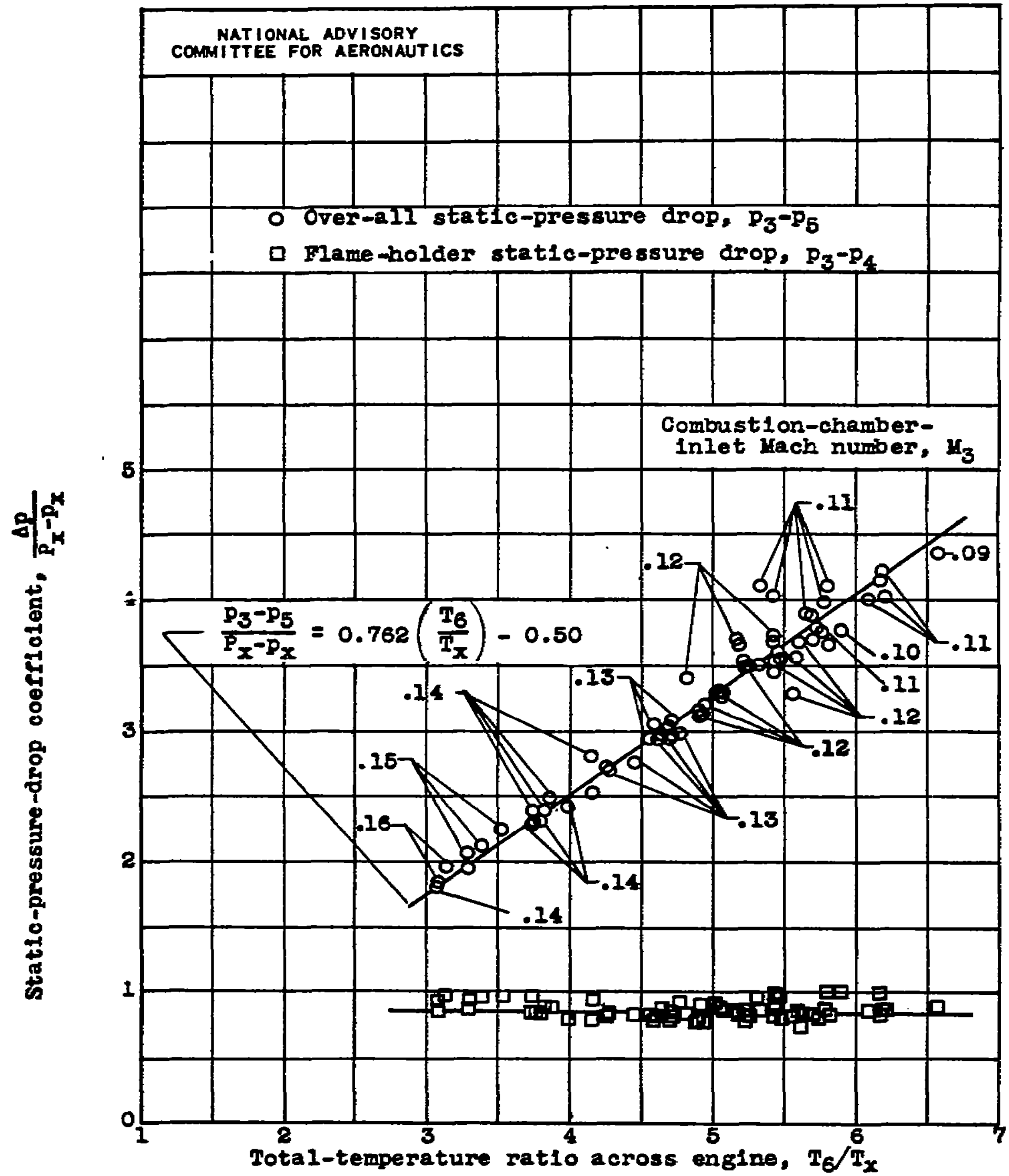
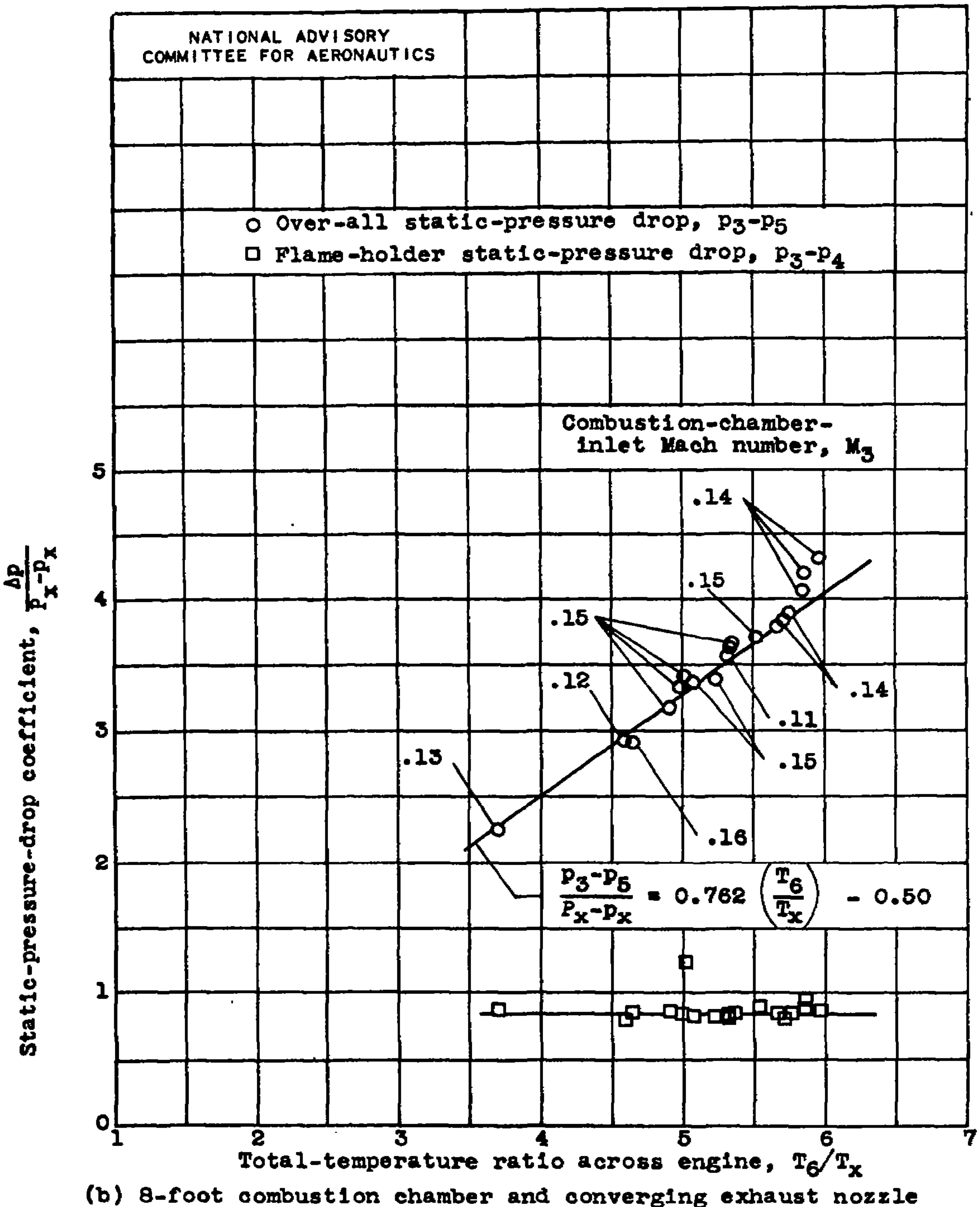


Figure 4. - Theoretical variation of combustion static-pressure-drop coefficient $\frac{p_3 - p_5}{2q_3}$ with $\mu^2 \left(\frac{R_5}{R_3}\frac{T_5}{T_3}\right)$ for various values of combustion-chamber-inlet Mach number M_3 with γ_3 = 1.4 and γ_5 = 1.3.



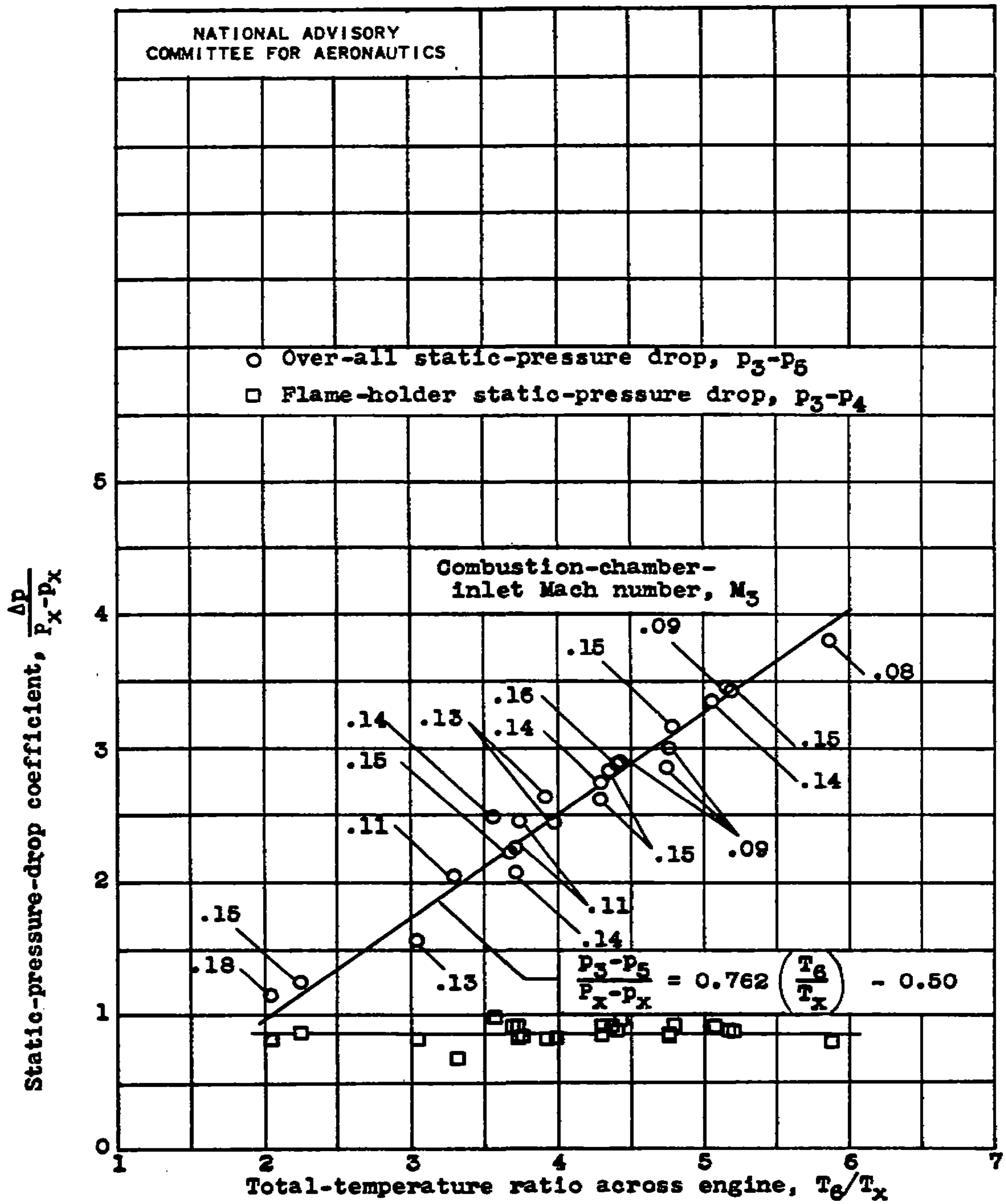
(a) 8-foot combustion chamber and converging exhaust nozzle with 15-inch exit diameter.

Figure 5. - Variation of flame-holder and over-all combustion-chamber static-pressure-drop coefficients with total-temperature ratio across engine for 20-inch ram jet.



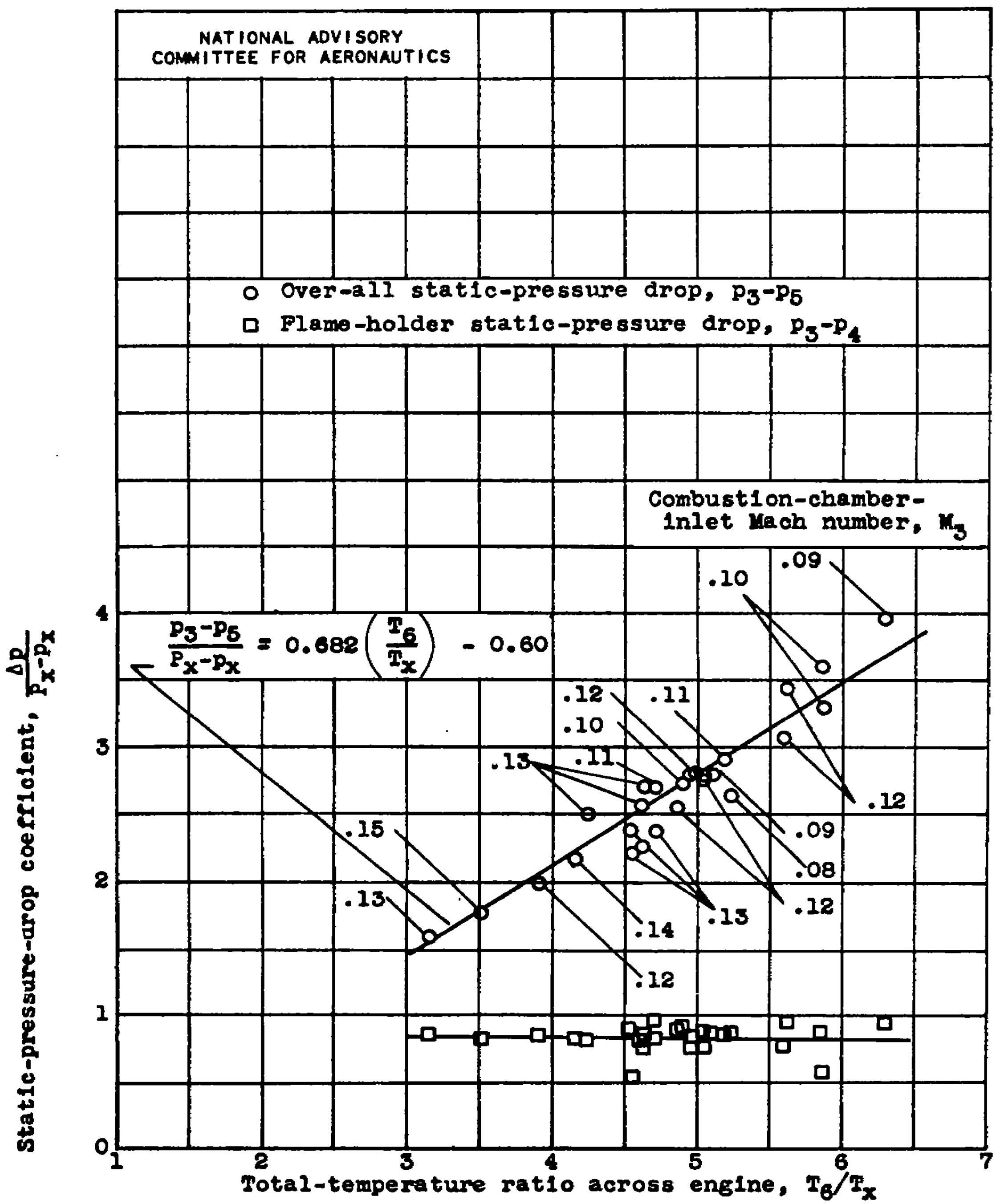
(b) 8-foot combustion chamber and converging exhaust nozzle with 17-inch exit diameter.

Figure 5. - Continued. Variation of flame-holder and over-all combustion-chamber static-pressure-drop coefficients with total-temperature ratio across engine for 20-inch ram jet.



(c) 8-foot combustion chamber and converging exhaust nozzle with $18\frac{3}{4}$-inch exit diameter.

Figure 5. - Continued. Variation of flame-holder and over-all combustion-chamber static-pressure-drop coefficients with total-temperature ratio across engine for 20-inch ram jet.



(d) 5-foot combustion chamber and converging exhaust nozzle with 15-inch exit diameter.

Figure 5. - Continued. Variation of flame-holder and over-all combustion-chamber static-pressure-drop coefficients with total-temperature ratio across engine for 20-inch ram jet.

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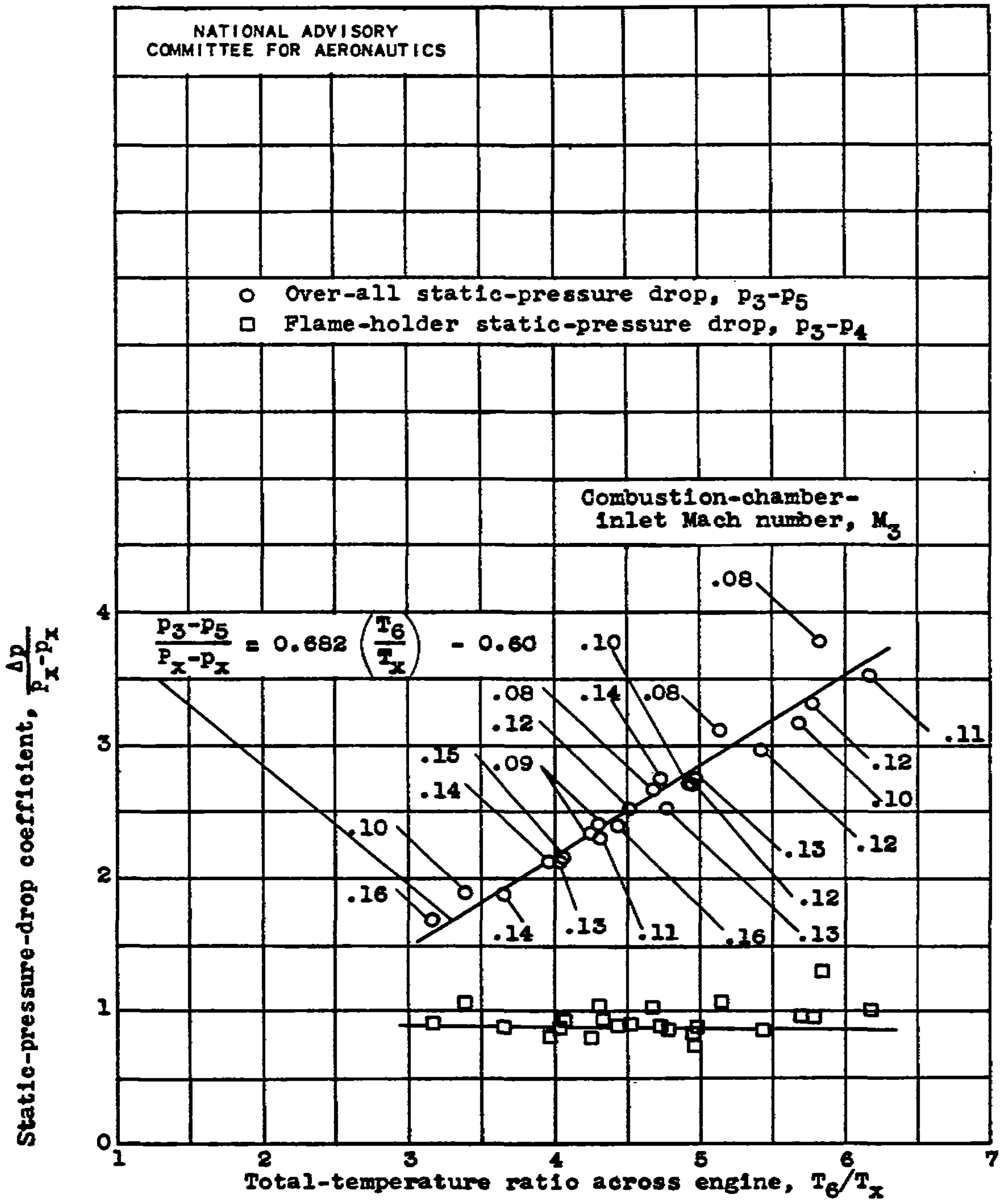
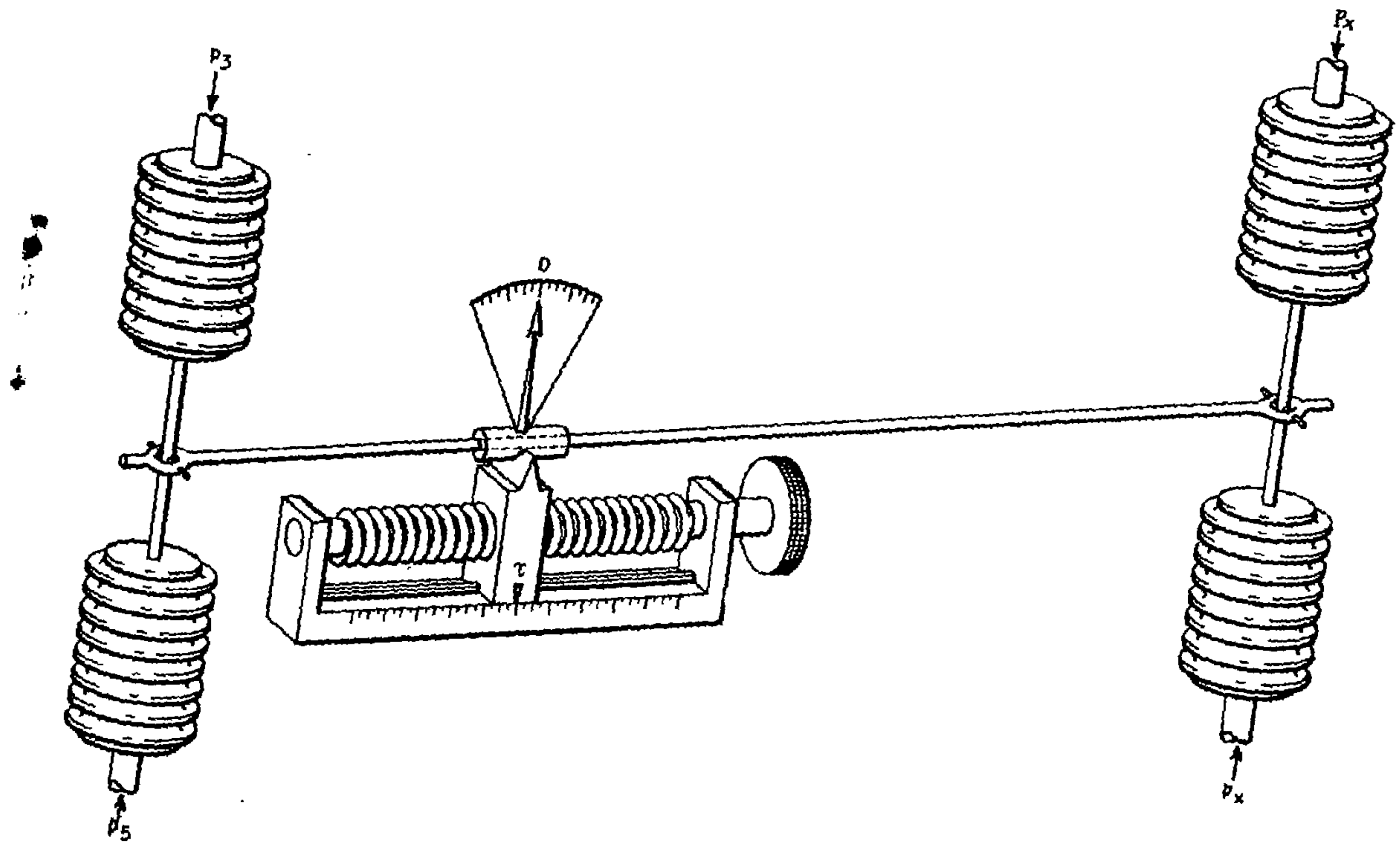


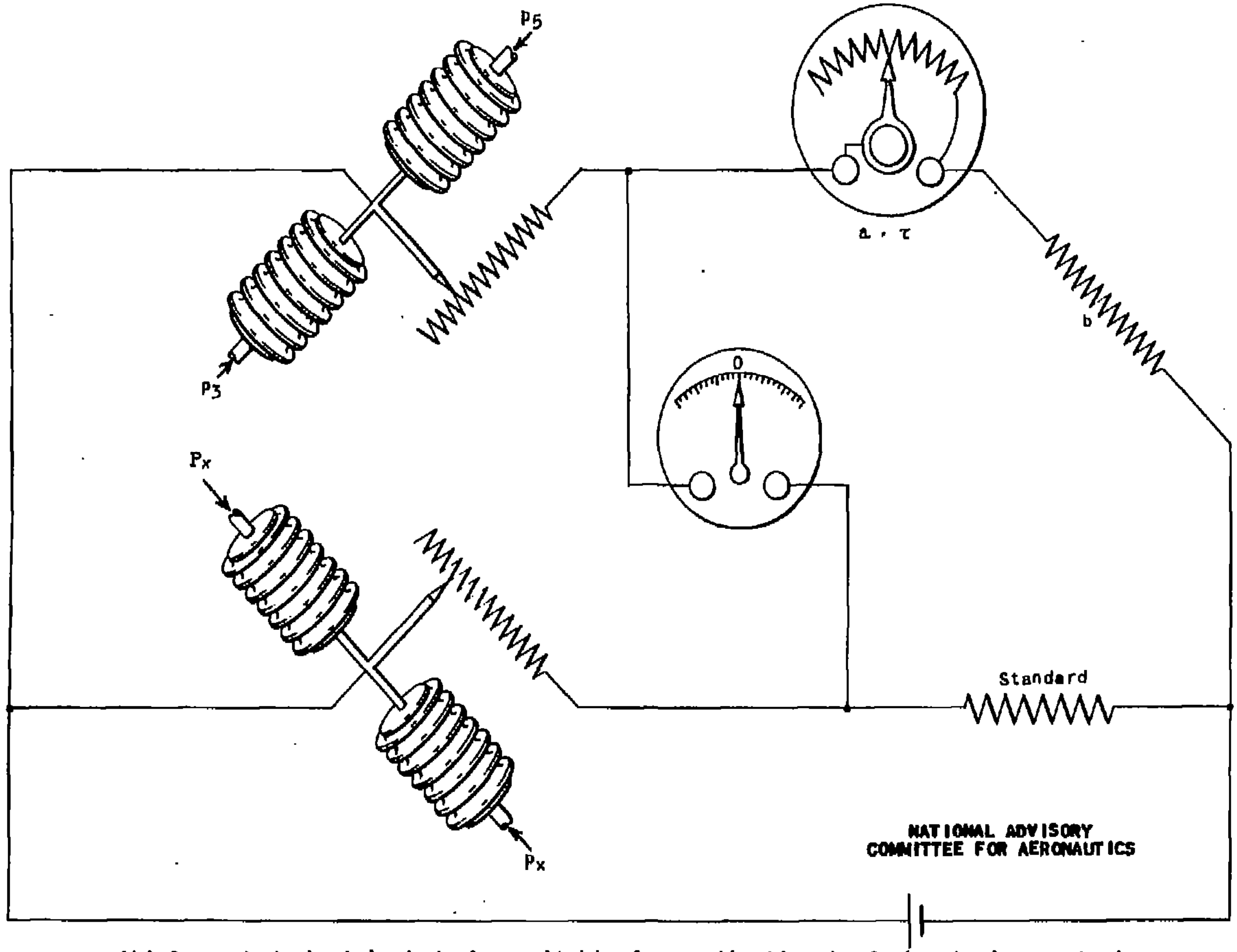
(e) 5-foot combustion chamber and converging exhaust nozzle with 17-inch exit diameter.

Figure 5. - Concluded. Variation of flame-holder and over-all combustion-chamber static-pressure-drop coefficients with total-temperature ratio across engine for 20-inch ram jet.



(a) Suggested mechanical design suitable for application to fuel-metering control.

Figure 6. - τ-meter based on combustion-chamber static-pressure-drop method of determining τ.



(b) Suggested electrical design suitable for application to fuel-metering control.

Figure 6. - Concluded. τ-meter based on combustion-chamber static-pressure-drop method of determining τ.

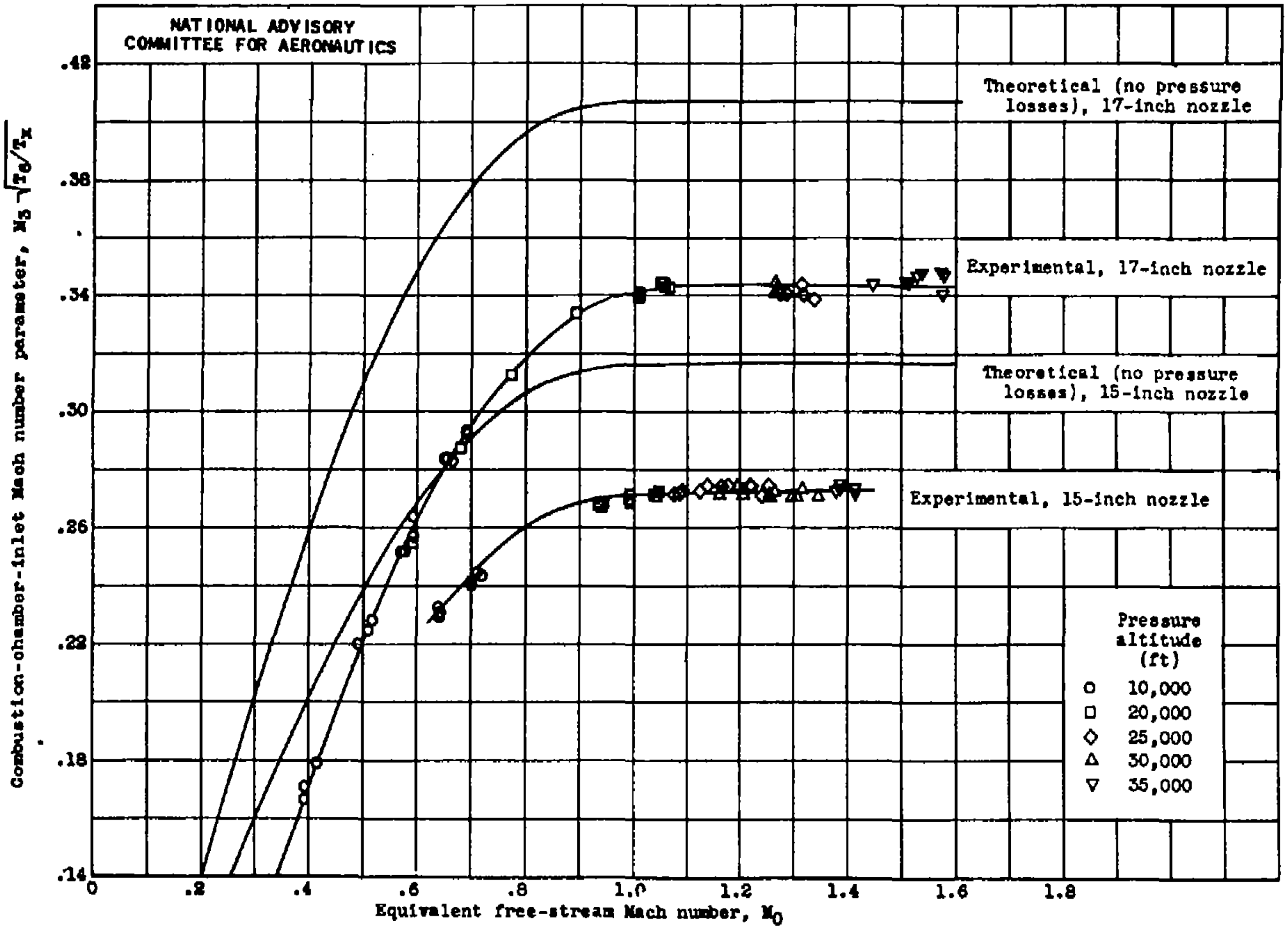
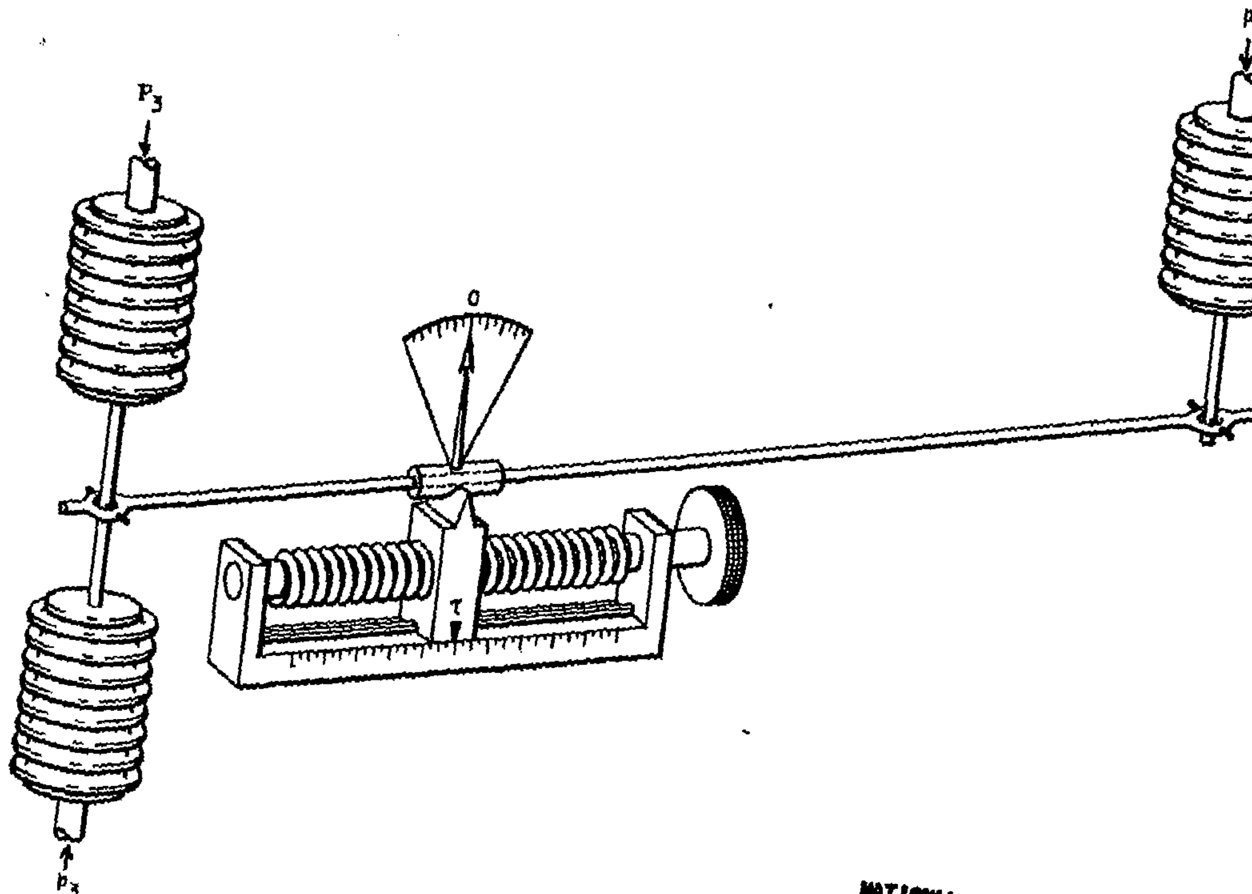


Figure 7. - Variation of combustion-chamber-inlet Mach number parameter $M_3\sqrt{T_6/T_x}$ with equivalent free-stream Mach number M_0 for 20-inch ram jet with 8-foot combustion chamber.



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(a) Suggested mechanical design suitable for application to fuel-metering control.

Figure 8. - τ-meter based on combustion-chamber-inlet Mach number method of determining τ.

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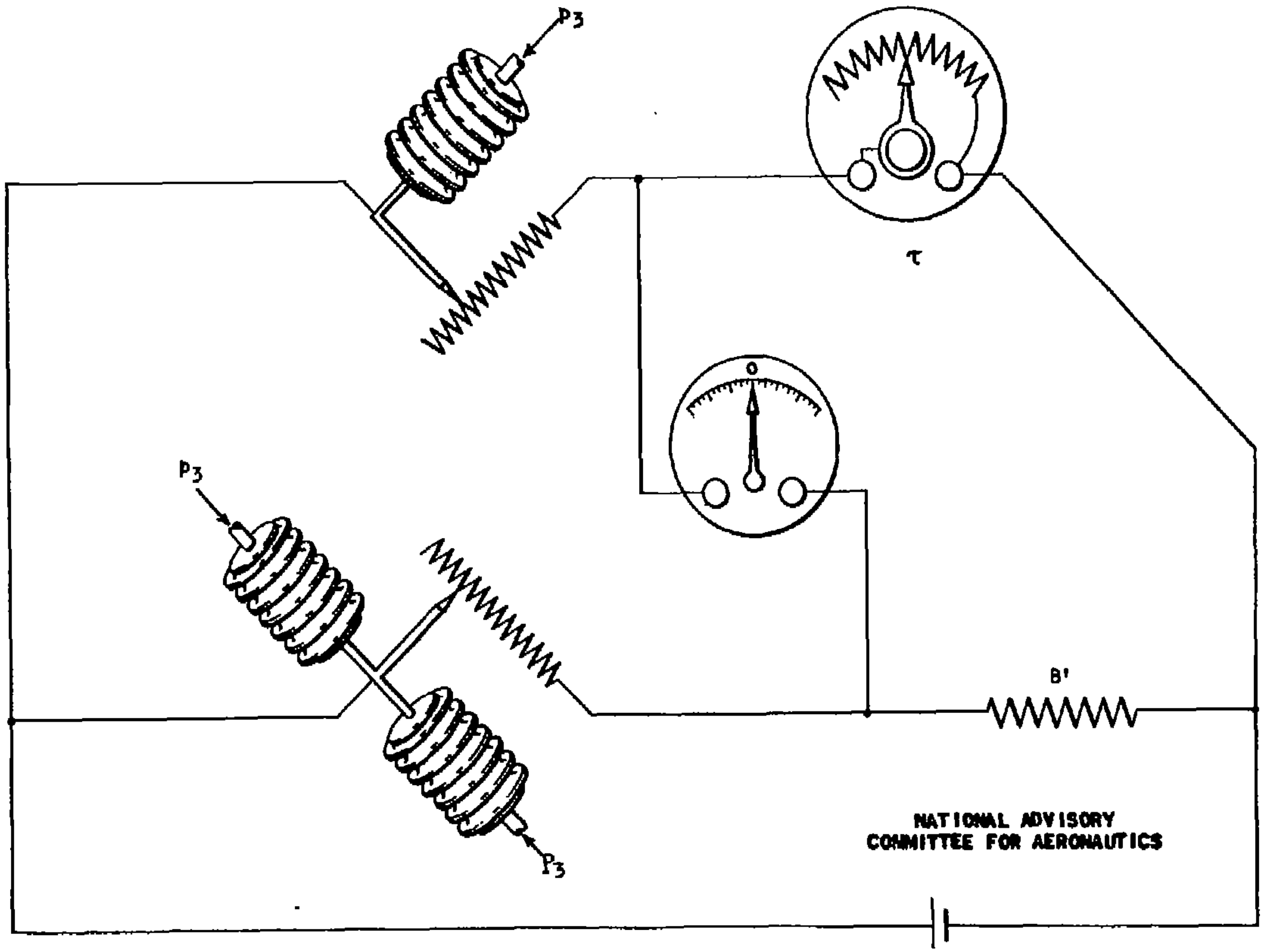

(b) Suggested electrical design suitable for application to fuel-metering control.

Figure 8. - Concluded. τ-meter based on combustion-chamber-inlet Mach number method of determining τ.

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